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Transition Prediction and Design Philosophy for Hybrid Laminar Flow Control for Military Aircraft

by
M. Maina and P. W. C. Wong
Aircraft Research Association,
Manton Lane, Bedford, MK41 7PF, UK

Abstract

An investigation into the application of transition prediction methods for hybrid laminar flow control for military aircraft has been carried out. Linear stability theory and the 'e^n' criterion are commonly used for predicting the onset of transition. Although a great deal of experience has been gained in their use over the years, there are still issues involved in the use of these methods which affect the accuracy of transition prediction. The investigation has shown the importance of the inclusion of negatively oriented oblique waves in the calculation of the N-factors. The effect of these waves on the predicted N-factor values and hence transition onset has been shown to be significant and cannot be ignored. These effects are more marked for the combination of adverse pressure gradients and high angles of sweep which are relevant to military aircraft.

A parametric study has been carried out to investigate the effect of surface suction or cooling on transition and to determine the appropriate suction quantities or cooling rates required to suppress the various instability modes in order to delay transition for military aircraft. Trends in the possible extents of laminar flow achievable for different pressure distributions and flow conditions with various suction and/or cooling distributions are shown and their implications for design are discussed.

Notation

c	Aerofoil or streamwise chord.
CF	Crossflow.
C_p	Pressure coefficient.
C_Q	Total suction quantity.
M_∞	Freestream Mach number.
n	N-factor in linear stability analysis.
R_c	Reynolds number based on chord normal to leading edge.
Rc	Reynolds number based on streamwise velocity and chord.
$\bar{R}$	Attachment line momentum thickness Reynolds number.
T	Temperature, in Kelvin.
T_w	Wall temperature.
T_∞	Ambient temperature.
TS	Tollmien-Schlichting.
V_s	Suction velocity, non-dimensionalised by freestream velocity.
x	Ordinate in streamwise direction.
I	Angle of incidence.
β^*, kz	Spanwise wavenumber. ($kz = \beta^* \times 10^{-3}$)
Λ	Sweep angle.
ψ, psi	Wave orientation.

Introduction

In recent years, there has been considerable interest in hybrid laminar flow control for aircraft applications, as this is seen as a promising technique for drag reduction and performance enhancement. This interest has been manifested in the initiation of a number of laminar flow research programmes which involve flight testing using various demonstration aircraft, for example, in Europe, the A320 laminar fin programme[1]. During the past few years, laminar flow research at the Aircraft Research Association (ARA) has been concerned with the evaluation of the suitability and accuracy of theoretical methods currently available to the UK aerospace industry for laminar flow aerofoil and wing design[2]. The methods have been applied in a parametric study which was carried out to investigate the effects of different pressure and suction distributions on transition location. This was initially aimed at the wing planform geometries and flow conditions typical of Airbus class civil transport aircraft[3,4].

Recently, the work has been extended to encompass pressure distributions and flow conditions more relevant to military aircraft. The wing planform geometries assumed in the analysis are typical of modern combat aircraft, with high leading edge sweep and thin wing sections. In addition to suction, the technique of surface cooling has been investigated as an

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alternative means of control. This work has been carried out under contract to BAE SYSTEMS and forms the subject of this paper.

The successful design of hybrid laminar flow aerofoils and wings requires methods for predicting the position of transition onset with sufficient accuracy to enable consistent comparison and which lead to satisfactory accuracy in drag prediction. A commonly used method for doing this is linear stability analysis. The paper describes this method and addresses some of the issues arising from its use. Results illustrating various conclusions drawn from the parametric study are presented and their implications for design are discussed.

Transition Prediction

On swept wings, transition from laminar to turbulent flow may be caused by various mechanisms, which depend mainly on streamwise pressure gradient, Reynolds number and sweep angle. These mechanisms are, principally, attachment line contamination, crossflow (CF) instability and Tollmien-Schlichting (TS) instability.

Attachment line contamination is associated with a source of gross disturbance, such as the wing body junction, from which undamped turbulence is convected along the leading edge, thus causing the entire wing to be turbulent. This phenomenon is essentially a 'bypass' mechanism and its occurrence may be predicted by means of the empirical criterion due to Poll and Paisley[(5)]. This criterion utilises the concept of the critical attachment line Reynolds number, $\bar{R}$, where $\bar{R}$ is a function of the leading edge velocity gradient and sweep. If this parameter exceeds 245 then the attachment line is deemed to be susceptible to gross contamination; if greater than 580 then TS type instability may occur.

CF instability is predominant in the leading edge region and is associated with the initial flow acceleration; it is strongly sweep dependent. The TS mode dominates further aft on the wing, particularly in adverse pressure gradient flows.

Simple empirical criteria may be used to predict transition due to either CF or TS instability, see for example Refs 6 and 7, but they are restricted in their range of applicability. Most of these criteria are formulated for incompressible flows only and none allows for the effects of boundary layer suction or cooling. Hence, these criteria were not considered suitable for the exercise described here.

In the UK aerospace industry, linear stability analysis methods, coupled with the 'e^n' criterion, are widely used for transition prediction. The theory behind such methods has been well documented elsewhere, see for example Ref 8, and therefore will not be repeated here. The method employed for the work described here is the spatial method, CoDS, due to Atkin[(9)]. A prerequisite for the use of these methods is the accurate determination of the velocity and temperature boundary layer profiles and their derivatives. This is achieved using a differential laminar boundary layer method subject to the assumption of semi-infinite, swept-tapered flow conditions. The swept-tapered flow assumption is reasonable provided that the isobars follow the wing generators. Such an assumption has been shown to apply to the outer wing region of a civil transport aircraft, see Ref 4.

The flow on modern combat aircraft wings is highly three-dimensional. Strictly, this would imply that a fully three-dimensional laminar boundary layer method should be used. However, consideration of the flow pattern on typical military wings has shown that on the forward part of the wing the isobar sweep approximates to the leading edge sweep. Since this is the region of the wing of interest in terms of the stability characteristics of the boundary layer, it has been assumed that the flow can be approximated by infinite yawed wing conditions, with the leading edge sweep being applied.

Linear stability analysis methods require the definition of an N-factor value at transition onset which may be used for either wind tunnel or flight conditions. Generally, these N-factor values are arrived at through correlation with experimental data, which leads to different values for different facilities due to varying freestream environment. The calculation of the N-factor requires the use of one of several integration strategies, the principal ones being:- constant wave orientation (ψ); constant spanwise wavenumber (β^*); constant wavelength (λ) and the envelope method. For fully three-dimensional flow, it is unclear which, if any, of these strategies models the physics of the problem most accurately. However, from kinematic wave theory it can be argued that, assuming infinite swept wing conditions, the constant β^* strategy has the greatest physical relevance for crossflow dominated flows. This integration strategy has been extensively evaluated and has been shown to produce consistent results[(10,11,12)]. For the work described in this paper, this is the integration strategy which has been adopted.

The choice of N-factor for transition onset is open to question, as it is dependent on the freestream environment. Values arrived at based on correlation with wind tunnel and flight test data range from approximately 6 to 20. The lower value is consistent with the correlation derived using experimental data from the DERA 8ft wind tunnel and given in Ref 10. Higher values are typical of flight conditions, reflecting the lower freestream turbulence levels encountered.

The use of linear stability analysis presupposes that the CF and TS instability mechanisms may be treated independently. However, for some flow conditions, it is possible that there would be some degree of interaction between the two modes and such flows may need to be analysed using a non-linear, non-local method such as PSE (Parabolised Stability Equations), e.g. Ref 13. Although such advanced methods are now becoming available, they have not yet been evaluated sufficiently for use in routine design applications and so do not form part of the analysis discussed here.

Given the integration strategy to be used, the accurate determination of the N-factor for a specific condition is clearly of prime importance. Since CF instability occurs very close to the leading edge for most practical wing flows, it is necessary to ensure sufficient streamwise grid resolution in order to, firstly, identify the neutral stability point correctly and, subsequently, to predict the correct trend in N-factor. Figs 1 and 2 show the effect on the predicted N-factor of having insufficient detail in the leading edge region. Referring to Fig 1, the CF modes correspond to the spanwise wavenumbers (kz) producing N-factors which reach a peak in the initial 10% chord; the modes downstream are due to TS instability. It can be seen for this case, that coarsening the streamwise grid has resulted in a reduction in both the disturbance growth rate and the maximum N-factor. This would imply a degree of uncertainty in the prediction of the transition position and the suction levels required to stabilise the flow. This is clearly an important issue for pressure data which is experimental in origin and, therefore, relatively sparse. For theoretical data as used here, although it is more likely that the grid resolution will be sufficiently fine, it is still important to ensure this.

An important point to note for the TS instability mode is that the most amplified disturbance may not be in the streamwise direction but may be oriented at some oblique angle. This is particular true for compressible flow conditions. It is essential that these waves be taken into account when carrying out the N-factor integration. In the previous work aimed at civil aircraft applications[4], the most unstable oblique waves were oriented in a positive sense, i.e. towards the crossflow direction. For the sweep angles relevant to the military applications considered here, it was apparent that the most unstable modes were negatively oriented. Fig 3 illustrates this point by showing a typical result in terms of the N-factor variation for a range of β^* values, where the most amplified mode for this case is $\beta^* = -900$ (kz = -0.9). The associated wave orientations are shown in Fig 4, where the most amplified mode, corresponding to $\psi \approx -60°$, is marked on the figure for clarity. The important point arising from the existence of the negative waves is that TS transition is predicted further forward than would have been the case if positively oriented waves only were considered, typically this forward movement is of the order of 10% chord, as can be seen in Fig 3. While the results are numerically correct, it is unclear whether they are physically feasible and more fundamental research would be required to ascertain this. Such research is beyond the scope of this paper.

Parametric Studies

The military wing geometries under consideration in this research programme tend to have relatively high leading edge sweep. This feature implies that transition would be likely to be caused at or close to the leading edge by either attachment line contamination or CF instability. This would need to be controlled in such a way as to suppress the instability sufficiently to delay transition to aft of the minimum pressure point. In the region downstream of this point TS instability would be the mechanism most likely to cause transition. This implies that the control methodology may be considered in two separate parts: firstly, to suppress CF instability and, secondly, to delay TS induced transition.

In previous work carried out at ARA[4], structural constraints for large civil transport aircraft dictated that boundary layer control, in the form of surface suction, could only be applied to the wing upper surface forward of the front spar, typically located at around 20% chord. For the military applications considered here, these constraints were relaxed. Although still confining the flow control to the wing upper surface, the use of both suction and cooling was investigated. The technique of surface cooling was included as it was considered that there might be structural advantages in practice compared to a suction system. The design of practical suction and cooling systems requires considerable research and is beyond the scope of this paper.

The previous work involved a systematic investigation of the appropriate types of pressure distributions for hybrid laminar flow wings for large civil transport aircraft[4]. This has been extended to military applications with the aim of establishing trends with sweep and Reynolds number which could provide guidance for hybrid laminar flow wing design.

Pressure distributions

In the work described in Ref 4, a series of pressure distributions were constructed which were relevant to cruise conditions for the aircraft types of interest. These pressure distributions were defined in two parts: first the initial gradient just downstream of the attachment line and second the 'rooftop' gradient. A range of leading edge and 'rooftop' gradients was covered as indicated by the sketch in Fig 5. The initial gradient is related to the leading edge geometry of the wing section, with the steeper gradient associated with a smaller leading edge radius. Conventional civil aircraft wings would correspond to the less steep initial gradients. The smaller leading edge radii, and therefore steep initial gradients, are typical of combat aircraft wings or unconventional sections, such as the Pfenninger type. The smaller leading edge radii of the military aircraft wings are the result of the significantly thinner wing sections used compared to civil wing sections. In certain flow conditions, this type of section geometry may lead to the development of a pressure distribution that has a high suction peak in the leading edge region of the wing upper surface. For some military wings, this occurs at cruise conditions with the peak height varying across the span. The adverse pressure gradient downstream of the suction peak may lead to the occurrence of a laminar separation bubble, which in turn would cause forward transition.

Given the section geometry typical of military wings, for the purposes of the parametric study, the initial pressure gradient was restricted to the steepest case studied in Ref 4. This was followed by a mildly adverse 'rooftop' gradient, which may be an appropriate type of pressure distribution for cruise conditions. Previous work[3] has shown that such pressure distributions are beneficial for aircraft performance, in terms of lift to drag ratio. Additionally, a series of pressure distributions was constructed with a range of leading edge suction peaks of various heights, see for example Fig 6, which were considered to be relevant to military aircraft wings.

The parameters considered in the exercise were leading edge sweep, assumed to range between 40° and 60°, and Reynolds number. The Mach number range, 0.8 – 0.9, was assumed to be typical of subsonic cruise conditions for modern combat aircraft. Previous work has shown that transition location is relatively insensitive to Mach number in this range.

Attachment line contamination

As noted above, the likelihood of the occurrence of attachment line contamination may be assessed by applying the criterion due to Poll and Paisley[5]. Fig 7 shows the variation of $\bar{R}$ with Reynolds number based on chord normal to sweep for the range of sweep angles considered. Also shown is the critical value, $\bar{R}$ = 245, above which the attachment line is susceptible to gross contamination. This would imply that attachment line contamination needs to be controlled via a device, such as a Gaster 'bump', or suction, which would eliminate the propagation of disturbances emanating in, for example, the fuselage boundary layer. Ref 14 discusses the use of suction to relaminarise a turbulent attachment line and defines the quantities required. For the type of wing sections considered here which have small leading edge radii and, hence, steep initial pressure gradients, it can be seen that the values are well below the neutral stability value, $\bar{R}$ = 580, indicating that the attachment line is not susceptible to TS type instability.

Crossflow instability

To suppress CF instability sufficiently to delay transition it is necessary to apply boundary layer control so as to reduce the predicted N-factors below a specified value. As an example of this, Figs 1 and 3 show N-factors for the same flow condition, without and with suction, respectively. The exercise sought to establish the minimum amount of control necessary for the sweep and Reynolds number ranges of interest.

Given the uncertainty mentioned earlier regarding the N-factor for transition onset, an understanding of the sensitivity of the minimum suction requirement to assumed N-factor was gained by considering a number of different values. The chosen N-factors were 8, 10 and 15, which should cover the expected values found in wind tunnel facilities and flight conditions. To illustrate this sensitivity, Fig 8 shows the variation of the minimum required suction velocity, V_s, with Reynolds number based on chord normal to the leading edge, R_c, for $\Lambda = 40°$. Similar variation can be seen at other sweep angles. CF instability is strongly dependent on sweep as is shown in Fig 9. This figure also illustrates the point that the minimum suction requirement tends to an asymptote as Reynolds number increases. Here n = 10 has been assumed as an appropriate value for transition onset and will be used throughout the analysis presented here. This is lower than some results quoted for flight conditions but is considered a reasonably conservative value for use in design studies. The use of the normal chord Reynolds number instead of the freestream Reynolds number is considered more appropriate for identifying the effect of sweep, as it implies effectively the same wing chord for the different geometric conditions. From these results, interpolation or extrapolation to a particular condition would be possible.

The results discussed above are based on applying suction over a region of the wing surface covering the initial flow acceleration and including the minimum pressure point for the simple pressure distribution type shown in Fig 5, in this case 8% chord. However, it would also be possible to concentrate the suction over smaller chordwise extents. The use of shorter suction panels and, therefore, smaller plenum chambers may be beneficial for suction system design. This type of arrangement may be necessary when the effect of pressure gradient on suction velocity is taken into account. Fig 10 shows the minimum suction velocity required to constrain the CF N-factor to below 10 for a range of suction extents. The required suction velocity remains constant until the suction extent is less than 2.5% chord, when a sharp increase is observed. Although the suction velocity increases with reduction in panel length, the total suction quantity, C_Q, reduces, see Fig 11. The value of C_Q may be more relevant in terms of suction system design, as it is directly related to the pump drag. This implies that applying a higher suction velocity over a shorter suction panel may be beneficial when total drag is considered.

An alternative technique to suction for controlling instability is surface cooling. When this was investigated for the same cases described above, it was found that the required cooling rates were very high. As an example, Fig 12 shows the variation in maximum CF N-factor with wall temperature at two Reynolds numbers. It can be seen that to reduce the N-factor to below 10 as for suction, it is necessary to apply cooling rates, T_w/T_∞, < 0.5, which is probably impractical.

As commented earlier, military aircraft wings may have pressure distributions with a leading edge suction peak which may induce separation. The likelihood of a separation depends on the severity of the adverse pressure gradient downstream of the peak. The pressure distributions shown in Fig 6 were used to establish the suction quantity required to suppress both the separation and CF instability. Here suction was applied over the initial 5% chord covering the region of the suction peak. The results are shown in Fig 13 in terms of the minimum suction requirement for both separation control and suppression of CF instability, assuming n = 10. Clearly, for a given Reynolds number, there is a maximum peak height attainable before separation is predicted and below which suction is required for CF instability control only. Above this maximum, it appears that suction is a viable means of separation control, although it should be noted that, for the highest suction peak case, the suction velocities required are very high, and this may be unrealistic in practical terms.

Tollmien-Schlichting instability

Having delayed the onset of transition to aft of the minimum pressure point, TS instability becomes the dominant mode. Transition in this case depends on the 'rooftop' pressure gradient and Reynolds number. In previous work[3], it was shown that, as Reynolds number increases, the predicted TS transition position becomes increasingly insensitive to pressure gradient. In contrast to CF instability, TS transition is independent of sweep. This is illustrated by Fig 14 which shows the transition location predicted assuming n = 10 for cases for which CF instability has been suppressed by the application of suction over the initial 8% chord.

For the types of pressure distribution considered here, in principle, boundary layer control may be used to delay TS transition to any specified position, provided this point is forward of a shock or recompression. Either suction or cooling may be used to achieve this, since both techniques are known to be effective for TS instability. Considering Fig 3 as a typical example of the N-factor variation for these types of pressure distribution, it can be seen that the suction applied to suppress CF instability is sufficient to move the TS neutral stability point to the end of the suction panel. It is therefore unnecessary to control TS instability with a suction or cooling panel immediately downstream; instead it is more efficient to place the panel further aft, allowing the instability to amplify until just below the value assumed for transition before control is applied.

As an illustration of the effectiveness of the two control techniques, Figs 15 and 16 show the variations in predicted transition onset position for suction and cooling respectively. For this case, which was chosen near the upper limit of TS sensitivity to the severity of the adverse pressure gradient, control is applied downstream of 15% chord. The results cover a range of suction velocities and cooling rates applied over different extents for two Reynolds numbers. As with CF instability control, there is a tendency for the variation in transition position to become independent of Reynolds number as the amount of control is increased. Clearly it is more effective to increase the extent of control than to increase the suction or cooling rate. It is worth noting that, in comparison with CF instability control, the suction velocities required are considerably reduced. The use of cooling may be a feasible technique for TS control as the cooling rates are not excessive; in this case, $T_w/T_\infty \approx 0.8$ has been shown to be sufficient for suppressing the instability and achieving a significant extent of laminar flow.

These results illustrate the use of a control methodology whereby discrete panels are used to suppress CF instability initially and subsequently to delay transition due to TS instability. For the example shown, suction can be applied over the initial 8% chord and then from 15% to 35% to delay transition to $x/c > 0.4$ for a streamwise Reynolds number of 40×10^6. For a less severe adverse pressure gradient, this methodology may be extended to encompass two or more discrete control panels for TS instability control, for instance 20 – 30% and 40 – 50%. Figs 17 and 18 illustrate this control strategy by showing the predicted N-factors for a mildly adverse pressure gradient case with $\Lambda = 40°$, $Re = 40 \times 10^6$. In these cases, suction has been applied over a single panel, placed at 20 – 30% chord (Fig 17), and over two separate panels, with the second placed at 40 – 50% chord (Fig18). The same suction velocity has been used in both cases.

Design Implications

The relatively high leading edge sweep typical of modern combat aircraft wings implies that the most likely cause of forward transition is CF instability. As a result of the parametric study, trends have been established for the minimum suction velocity required to suppress this instability sufficiently to delay transition. The asymptotic behaviour of this quantity with increasing Reynolds number implies that a maximum suction velocity can be defined for a given geometric configuration. This would allow the maintenance of laminar flow onto the inboard wing without incurring the penalty of excessively high suction rates.

The thin wing sections, with their correspondingly small leading edge radii, lead to pressure distributions with steep leading edge gradients. This has two important benefits. Firstly, the attachment line Reynolds number can be kept to a sufficiently low value that leading edge contamination may be avoided, either by using a Gaster 'bump' or applying suction. Secondly, the steep gradient helps to reduce CF instability and restrict it to a relatively short chordwise distance, thereby permitting the use of small suction panels in the leading edge region. On the other hand, this type of leading edge geometry may produce pressure distributions with a suction peak close to the leading edge at cruise conditions. If this suction peak is sufficiently high, the steep adverse pressure gradient downstream may precipitate laminar separation. It has been shown that suction can be used to prevent the occurrence of this separation. However, for low Reynolds numbers the required suction velocity increases rapidly with peak height thus implying a maximum peak height for practical applications. Therefore, careful design of the leading edge geometry is required to achieve the right balance in terms of the trade-off between the steep favourable initial gradient and the likelihood of causing laminar separation.

Transition due to TS instability has been shown to be independent of sweep. In addition, in previous work, it has been observed that TS instability becomes insensitive to the steepness of the adverse pressure gradient at higher Reynolds numbers. As a consequence, it is more effective to increase the chordwise extent over which boundary layer control acts than to increase the level of suction or cooling. This enables the application of a control technique whereby discrete suction or cooling panels are used. The suction velocities are small, typically an order of magnitude lower than that required for CF control. The exercise has established that cooling is a viable control technique, particularly for moderately adverse pressure gradients.

It has been shown that the required suction levels are sensitive to the value of the N-factor assumed for transition onset. This is particularly important for CF instability control, since an underestimate in the suction quantity required would lead to a total loss of laminar flow. For TS instability control, sensitivity to N-factor in terms of chordwise extent of laminar flow is generally less as the amplification rates for the disturbance waves tend to be high. However, it is important that all unstable modes are predicted as failure to do so may result in an over-prediction of transition position by more than 10% chord. This sensitivity to N-factor must be borne in mind as inaccurate prediction of the transition onset position will result in misleading estimates of wing performance, as has been illustrated in Ref 3 for hybrid laminar flow aerofoils.

Concluding Remarks

a) It has been shown to be possible to use boundary layer control to delay transition onset for the types of pressure distributions relevant to modern combat aircraft wings. For these pressure distributions, transition would occur close to the leading edge due to either attachment line contamination or CF instability. For the leading edge geometries of interest, attachment line contamination may be readily avoided by means of suction or a device such as a Gaster 'bump'. The use of suction to suppress CF instability sufficiently to allow the onset of transition to move aft of the minimum pressure point, permits the use of either suction or cooling to control TS instability.

b) Clear trends with leading edge sweep angle and Reynolds number have been established in the variation of the minimum suction velocity required to suppress CF instability. It has been shown that the required suction quantity tends to become asymptotic at higher Reynolds number. It is beneficial in terms of total suction requirement to consider the use of short porous panels with a higher concentration of suction. The parametric study has provided sufficient information to enable the required suction quantity for a given leading edge pressure distribution to be established.

c) Suction has been shown to be a viable means of avoiding laminar separation resulting from the steep adverse pressure gradient downstream of a suction peak in the wing leading edge region. The results have shown that there may be a maximum peak height for which laminar separation can be prevented with the level of suction achievable in practice.

d) Transition due to TS instability has been shown to be independent of sweep. The concept of using separate suction panels or cooling strips for controlling TS instability has been shown to be feasible. The suction quantities required are significantly lower than those required to suppress CF instability. For a specified chordwise extent, the suction or cooling rates become asymptotic at higher Reynolds numbers. Consequently, it is more effective to increase the chordwise extent over which control is applied than to increase the suction or cooling rate. The effectiveness of using a succession of discrete control panels has been demonstrated.

Acknowledgement

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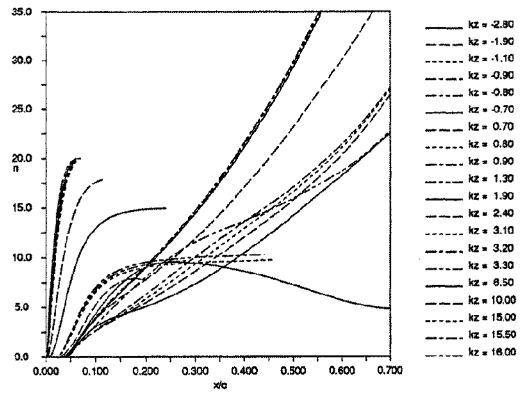

Fig 1 N–factors – Standard Point Distribution

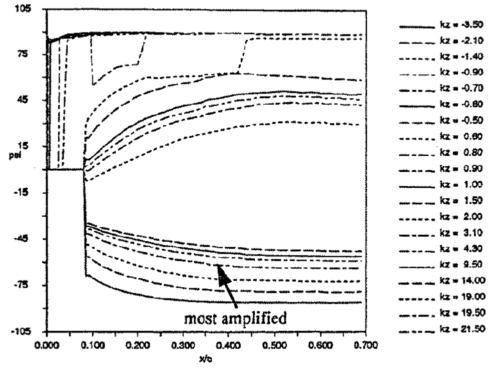

Fig 4 Wave Orientation – ψ

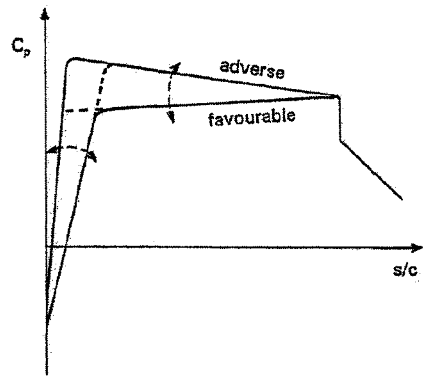

Fig 5 Sketch of Pressure Distributions

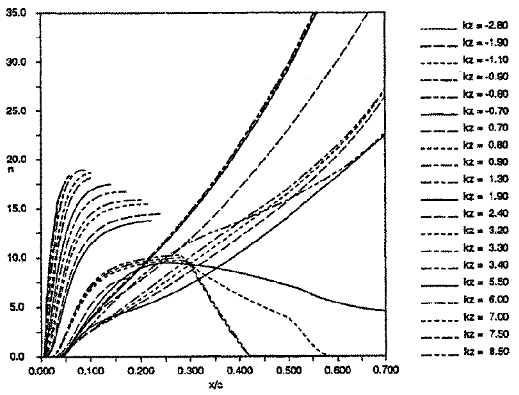

Fig 2 N–factors – Coarse Point Distribution

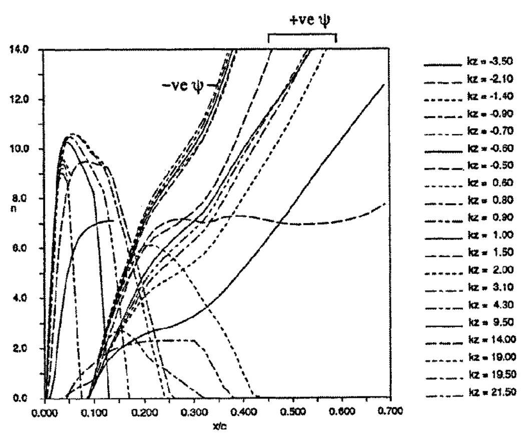

Fig 3 N–factor Variation

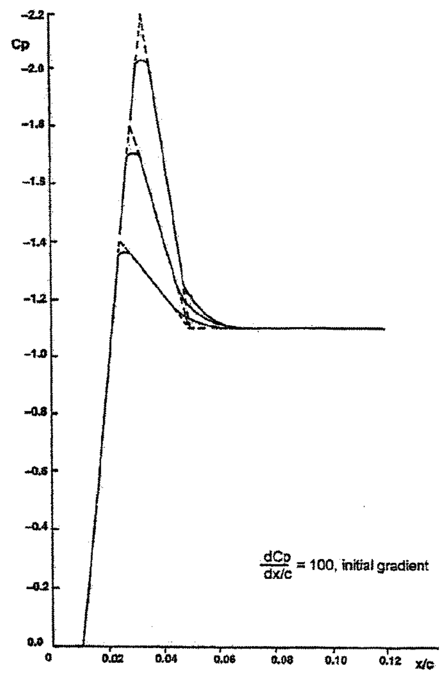

Fig 6 Pressure Distributions with a Range of Suction Peaks

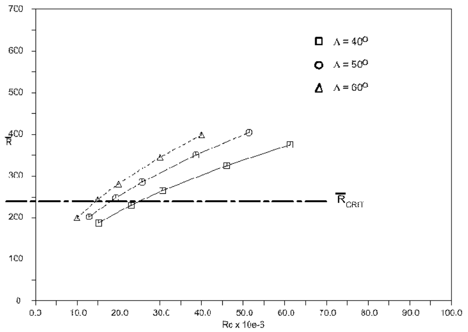

Fig 7 Variation of Attachment Line Reynolds Number

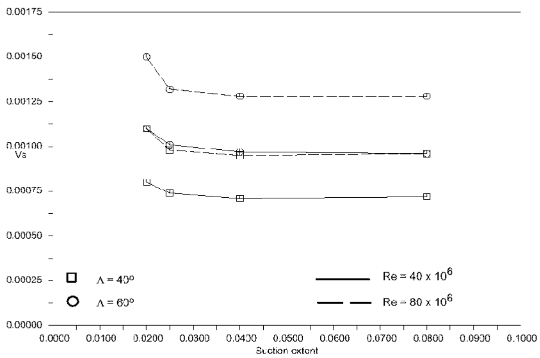

Fig 10 Leading Edge Suction Velocity

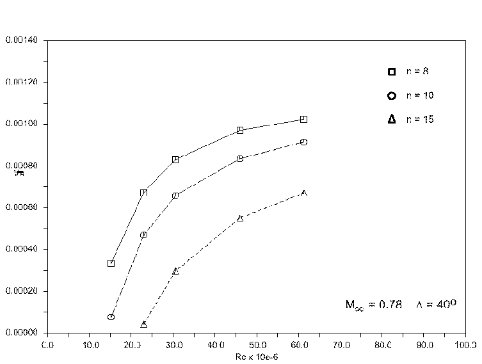

Fig 8 Effect of N–factor on $V_{s_{min}}$ for CF Instability

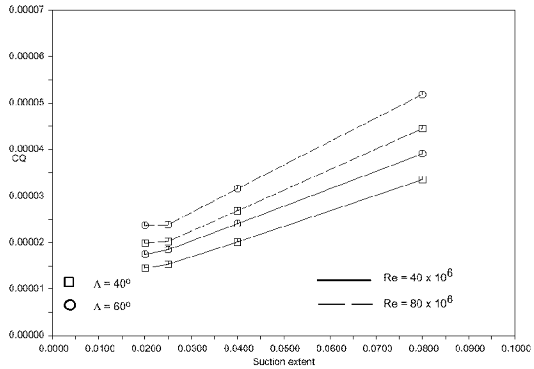

Fig 11 Leading Edge Suction Quantity, C_Q

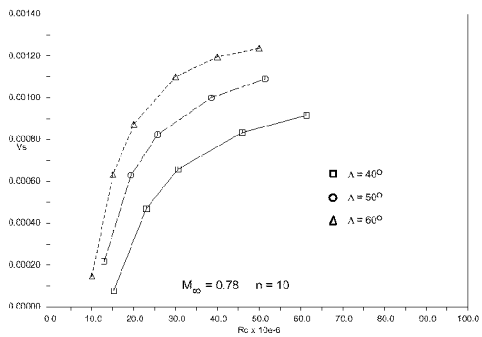

Fig 9 Effect of Sweep on $V_{s_{min}}$ for CF Instability

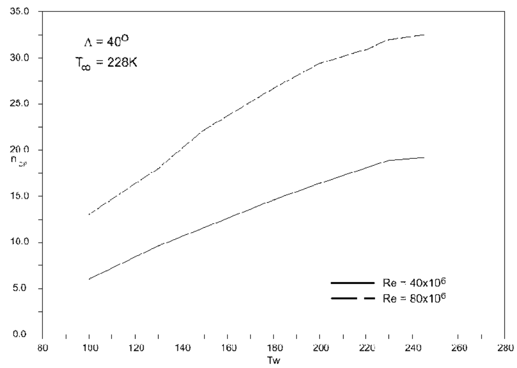

Fig 12 Effect of Cooling on CF N–factor

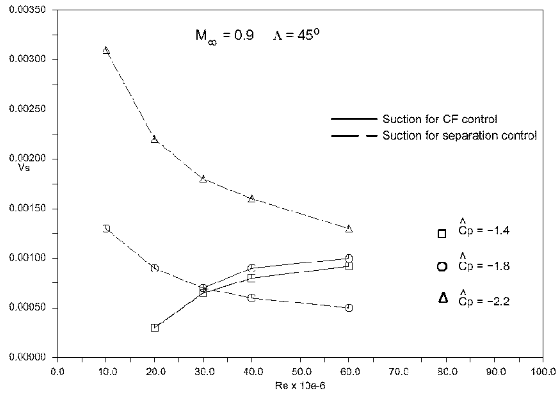

Fig 13 Suction Variation for CF and Separation Control

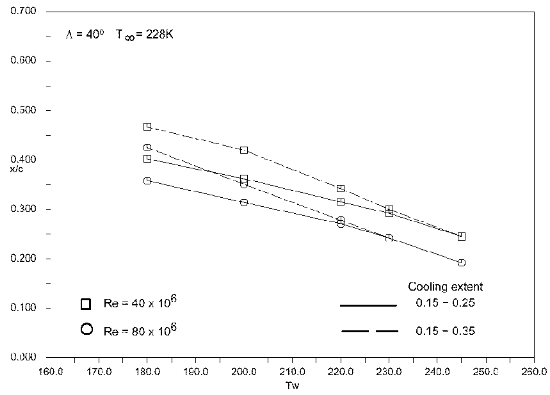

Fig 16 Effect of Cooling on TS Transition

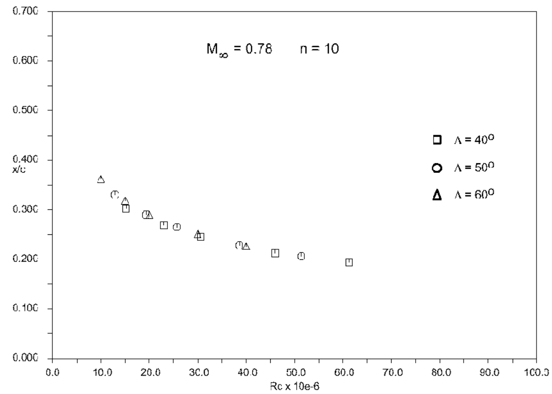

Fig 14 Effect of Sweep on TS Transition Location

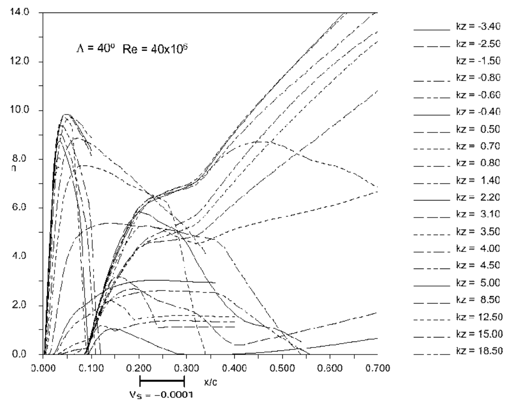

Fig 17 N–factor variation for TS Control
Suction Extent x/c = 0.2 0.3

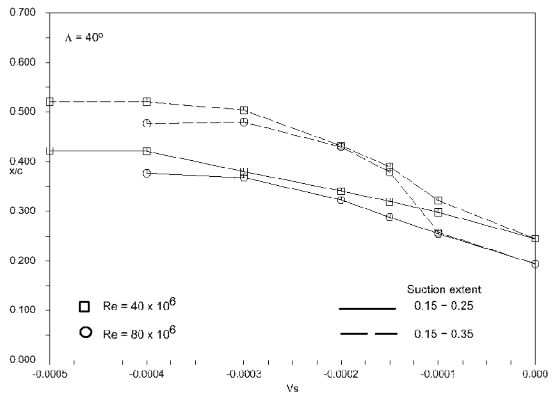

Fig 15 Effect of Suction on TS Transition

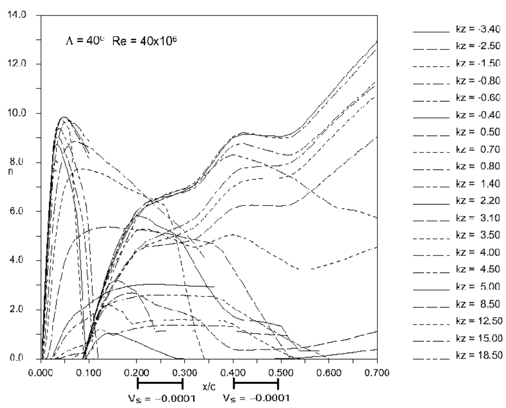

Fig 18 N–factor variation for TS Control
Suction Extent x/c = 0.2 0.3, 0.4 0.5